SIR-C/X-SAR Free Flyer Engineering Concept

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Abstract

After the Space Radar Laboratory (SRL)-1 and -2 missions, there is an opportunity to integrate the Spaceborne Imaging Radar-C/X-Band Synthetic Aperture Radar (SIR- C/X-SAR) instrument with spacecraft bus subsystems for a 1998 launch. Integration is accomplished using computer-aided design (CAD) systems and mass properties programs. The Free Flyer will be launched into a 400-km 57-deg-inclined circular orbit by the Shuttle, where it will remain for two years as a multifrequency multipolarization radar-observational platform. Its gross weight in orbit will be 12,707 kg and it will contain 1530 kg of fuel for three-axis inertial control and orbital adjustments consistent with the science objectives for the first 90 days. After on-orbit checkout, global mapping will be done with X-, I.-, and C-band frequencies and dual polarization. The next 45 days, it will regionally map with full polarization. A global topography phase follows, using repeat pass interferometery at L-band. The last 18 months will be spent doing differential interferometry observations, maneuvering the spacecraft to Earth events, and observing the resulting surface deformations.

1. Introduction

Spaceborne Imaging Radar-C/X-Band Synthetic Aperture Radar (SIR-C/X-SAR) is a joint U. S.-European mission scheduled to fly in April with a reflight in August 1994, The Shuttle-based flights will provide multiple wavelengths and polarized radar data to study Earth's ecosystems, climatic and geological processes, the hydrologic cycle, and ocean circulation. The system will allow scientists to make detailed studies of the Earth's surface on a global scale, including new measurements such as biomass (surface plant material) and soil moisture.

Nineteen geographic supersites of high scientific interest have been identified that will have high data-collection priority. These data will be used to correlate radar data with ground truth observations as a foundation for global mapping. The SIR-C/X-SAR mission will return 32 terabits of data–the equivalent of 20,000 encyclopedia volumes.

The follow-on, long-playing mission is the SIR-C/X-SAR Free Flyer. It promises greatly improved science with over 40 times the data volume, approaching the science return of the Earth Observing Satellite Synthetic Aperture Radar (EOS SAR) mission, but at a greatly reduced cost. Capitalizing on the large antenna structure, instrument electronics and spacecraft subsystems can be integrated into a single observational platform. Verified estimates show that a wraparound spacecraft can be done for a modest cost of about $110M. This includes spacecraft test and engineering, all spacecraft procurements, and subsystem developments.

The data return is extraordinary, producing nearly 1.5 terabits per day of three-frequency quad-polarization SAR data. After 107 days, 67 percent of the planet is mapped simultaneously in swaths up to the 57-deg latitude; 98 percent coverage can by obtained during the next 107 days. However, a preferred mapping mode may be dual-polarization with at least 15 min per orbit. All three frequencies can map 75 percent of the planet's land mass in 43 days and do a complete mapping in 90 days. More three-frequency mapping, using quad pol, can be done regionally over the next 45 days.

After this first 135-day mapping phase, the mission enters its second phase, a 56-day interferometric three-frequency mapping phase with single-pol data. This phase of the mission produces a global map with the same coverage limitations of the 57-deg orbit to 20-meter (m) accuracy, producing global U.S. Geological Survey (USGS) products, Several maneuvers are required to keep a cross-track baseline of 1500-2000 m.

The spacecraft is designed for a guaranteed two-year lifetime, so the remaining 18 months are devoted to differential interferornetry mapping, measuring centimeter-like changes to the baseline topography. Surface deformations due to earthquakes, volcanic eruptions, and glacial movements can all be detected.

The resolution is slightly degraded from the SIR-C/X-SAR mission and estimated to be within 50- by 50-m and 8 looks, SIR-C/X-SAR Shuttle missions are 30 m and 4 looks. Data products equivalent to the first mission can be generated if the Ground Data Processing Service (GDPS) is upgraded by a factor of 46, This may not be possible given anticipated growth of the present "thinking machines, " However, with the nonpolarimetric mapping standard, products can be produced regionally. Work is currently underway to examine the possibility of using even larger parallel processors, such as the CRAY T3D.

II. Mission Design

A. Design Constraints

The Free Flyer mission is constrained by Shuttle performance, available power, data handling, and cost. Although the two primary parameters affecting mission design are power and data handling, data processing is just as important. Either we must be able to process the data or reduce the amount of data acquired.

Rockwell International provided the performance for this mission (see Table 2-l). Shuttle performance can deliver 14,940 kg (29,370 lb) to 400 km at a 57-deg inclination, This altitude is selected to provide acceptable orbital lifetime and adequate radar performance, Although a positive margin is shown, the Free Flyer launch mass contains an 860-kg mass uncertainty. These estimates were derived from a complete system design compatible with mission requirements.

Table 2-1. Shuttle Imaging Radar-D Mission Launch Margin

Max. Payload Launch Capability	14,940 kg (32,928 lb)
Payload (SIR-C and propellant)	12,707 kg (28,006 lb)
STS Operator Reserve	192 kg (423 lb)
Launch Margin	2,041 kg (4,498 lb)

After achieving a 400-km orbit, the spacecraft is deployed using a Stabilized Payload Deployment System (SPDS). The SPDS is fully flight qualified and operational. Figure 2-1 depicts the SPDS lifting the payload in the vertical direction, translating it laterally, then rotating (rolling) over the side. The payload now is deployed, although alongside the orbiter. While still attached, first-stage deployment of the solar array can occur, rotating the solar array cannister into position and latching it. (An axial arrangement of the solar array is baselined. The width of the Shuttle cargo bay requires an additional deployment mechanism, This axial arrangement reduces gravity torques and enables pure couples for the attitude control system,) After the spacecraft is released, pyro devices are used to deploy the communications antenna.

Once the orbital checkout is complete, the Shuttle can separate and maneuver away safely (without plume impingement to the arrays). At this juncture, the spacecraft completes deployment of the solar array and begins the attitude acquisition and orbital trim maneuvers necessary to achieve the final operational orbit. The spacecraft nominal attitude will be rolled 35 deg and constantly pitched to maintain alignment to the velocity vector. The solar array flies edge-on to the direction of flight and rotates about one axis to "catch" the Sun's yearly progression. Power calculations have assumed the worst-case Sun orientation when the orbit is edge-on to the Sun or vice versa.

Spacecraft system design is highly interactive with mission design, especially since power and data handling are the two most important mission parameters. SIR-C, with three frequencies, quad polarization, and a full spacecraft, demands very high power, approximately 10 kW. The largest solar array available is being built for the EOS AM mission at about 6 kW. Adding panels to this array (extending the boom) does not defeat the qualified design and generates 8.25 kW of power. Other designs may also be available. Fortunately, this limit is sufficient to produce 15 min of dual pol L-, C-, and X-band SA'R data every orbit under the worst conditions.

Cruise power requirements of nearly 1100 W adversely affect the available power. This calculation assumes the instrument is turned off during cruise, except for the Command, Timing Telemetry Assembly (CTTA), so command and telemetry is always available. If this assumption holds and the Sun is at the worst-case edge-on time of the year, the SIR-C mode can be taking data only about 9 to 10 min out of the 93-rein orbit. By using only dual polarizations, longer data takes (up to 15 rein) per orbit can be achieved, Table 2-2 shows the progressively longer data takes available as instrument capability is further reduced. All cases were generated at the lowest operating voltage of 23 V and a full pulsewidth, with a Pulse Repetition Frequency (PRF) of 1738.

Table 2-2 also indicates other parameters that affect performance. Notice that between the first two cases, we have maintained a constant data rate from the instrument at 135 Mbps. This is achieved by running the SIR-C at 5 MHz and varying the swath width. Resolution is degraded, but still

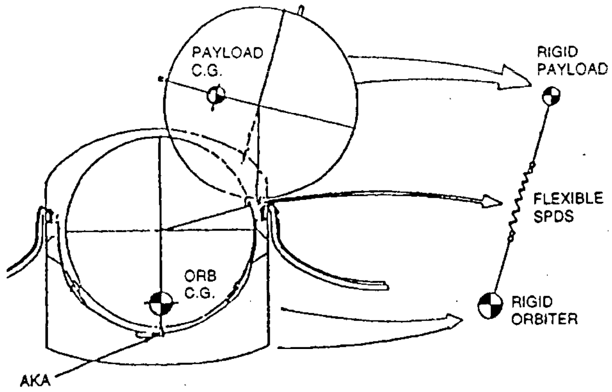

Figure 2-1. Orbiter, Payload, and SPDS Model Representation

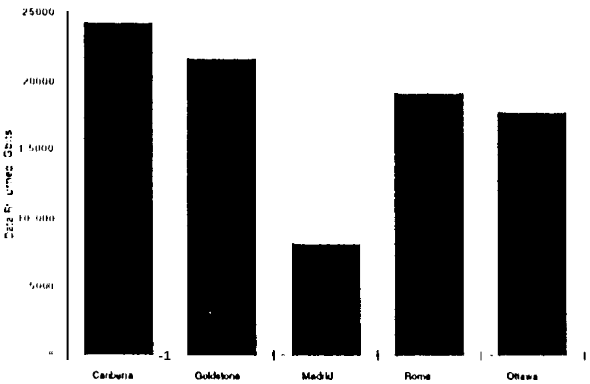

Figure 2-2. 43-Day Repeat Cycle, 135 Mbps of Data Returned

acceptable at 50 m. The X-SAR interface is still a constant 45 Mbps while SIR-C is at a constant 90 Mbps for L- and C-bands.

Dropping a whole frequency, as in case 3, allows the instrument to run at a quad-polarization at a 64-m swath width, By going to the dual mode in case 4, we simply increase the coverage by increasing the swath width to extended coverage mode. This allows for the most rapid mapping of 23 days. Case 5 is included since the longest mapping runs are about 30 min. For this case the data rate is reduced to 90 Mbps, and ground swaths do not have to be spliced.

B. Mapping Phase

With the instrument generating 135 Mbps, the spacecraft must be equipped to manage high data rates and high volumes. Earlier studies have shown that a Tracking and Data Relay Satellite System (TDRSS) link and a solid-state recording capacity of 320 Gbits are required to bring all the data down in the dual mode, e.g., mapping for at least 15 min around the descending node. However, since TDRSS capability is expensive to the spacecraft, the baseline solution of an X-band omni antenna with five stations has been assumed as an option. The X-band downlink operates at a constant 144 Mbps. Three Deep Space Network (DSN) stations—Goldstone, Madrid, and Canberra—are assumed. A station just outside Rome and a radar satellite station in Ottawa are also included. Figure 2-2 shows a typical data collection among the five stations. The amount of data downlinked depends on power available, on-board storage capability for a given repeat cycle, and the station mix.

Dual polarization reduces the power requirement and extends the possible coverage to 15 min per orbit. Figure 2-3 depicts this case. The larger swath width permits an optimum repeat cycle of 43 days. Although power is still limited, data rate begins to limit science at the 144-Mbps downlink because the dual L, C, X-SAR can capture 75 percent of the planet (land mass). This mode is ideal for mapping. Since quad-pol data is expensive in power, and the ability to process polarimetric data is very limited, full-polarized mapping is relegated to a third 45-day period for regional coverage.

Combinations of coverage can be constructed where we substitute full-polarized data takes for shorter strips (ground coverage) to be combined with the longer data takes in the dual mode. Also, the data constraint curve alone indicates that a downlink increase to 200 Mbps can significantly increase coverage to 85 percent during the first 43 days, leaving only a small amount for any subsequent mapping period.

C. Interferometry Phase (56 days)

After the mapping phase is complete, two maneuvers will trim the orbit slightly (about 1 km higher) to produce a 3-day orbital repeat track. These maneuvers are only about 1.3 m/s each.

Interferometric mapping is optimized for the L-band frequency in that a baseline separation of 2000 m is desired between passes. A 6-day period with orbital adjustments is required to maintain a baseline to produce a 140-km-wide (167 km along the equator) topological map. After the initial 6-day cycle, the orbit is moved 145 km along the equator–-allowing a 15-percent overlap to a new location–and desynchronized. This requires another 6.1 m/s and 3.7 days, then another 6-day repeat cycle. Each complete cycle takes 9.7 days and 8.6 m/s. The first five cycles cost 8.6 m/s each, while the sixth costs 2.5 m/s, for a total of 45.5 m/s and 56 days to fill in the space between the initial orbital swaths, approximately 8 deg apart at the equator.

Table 2-2. Power Calculations

Case	Thnc (Days)	Mode	Power* (W)	Data (Mbps)	Swath (km)	On-Time (rein)
1	107	L-Quad, C-Quad, X	1100/9800	135	29	10
2	43	L-Dual, C-Dual, X	1100/7500	135	64	15
3	43	L-Quad, X	1100/67001	135	64	18
4	23	L-Dual, X	1100/5000	135	140	24
5	43	L-Dual, X	1100/5000	90	64	27

*Cruise Power, 1100 W

D. Differential Interferometry Phase (18 months)

After obtaining a topological map at about 20-m accuracy, we can use the platform to observe events on the Earth which will produce surface deformation at the centimeter level. Any such event could occur so that a maximum of a 440-km shift in the ascending mode would be required. An average over the 18 months might then be 220 km, requiring 3.5 m/s to move and another 3.5 m/s to set the orbit in a new 3-day repeating ground track. This is a total of 7 m/s per event, and a maximum of one week. After the four days to move, it would take three days if the event were caught on the last orbit. An allocation of 50 m/s is necessary for these observations (at least seven).

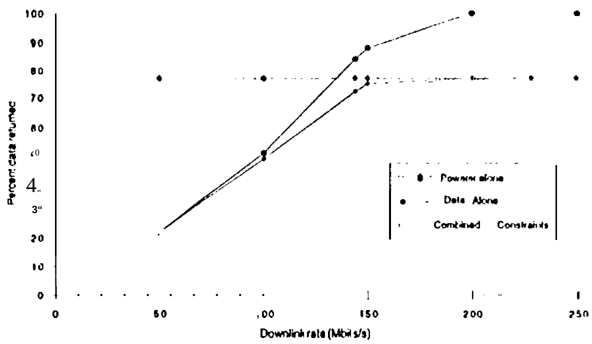

Figure 2-3. Percentage with Power and Data Volume, Separately and Combined

III. Systems Design

After the SRL-1 and -2 missions, the SIR-C/X-SAR instrument is available for upgrade to a free-flying satellite. The challenge to the system design is replacing the Shuttle as a resource and keeping its electrical and data interfaces within acceptable cost limitations.

Maintaining compatibility with the Shuttle means keeping the overall center of gravity nearly the same or aft of the present location. It also means not exceeding the trunnion load-carrying capability in any of the x-, y-, z-axes of the Shuttle, especially the keel trunnion x-axis which is near its limit.

Secondly, a shortened schedule (1998 launch) and the coordination required for a fully contributing partnership require as much modularity as possible, These two constraints, plus the sheer size of the spacecraft, lead to a rather interesting design.

First, the solar array *must be* out front in the direction of the velocity vector; it must be deployed edge-on to the direction of flight to minimize drag torques, which could arise from such a large area. Since thrusters must be placed at the structure's corners for accurate maneuverability, the array must be out in front far enough to minimize shading and plume impingement from coupled thrusters.

The front-mounted array is stowed longitudinally before release from the Shuttle. This is a result of the limited cargo bay size to accommodate the standard EOS design width. A mechanism and strong back is designed to secure the array to the Antenna Core Structure (ACS). The Galileo spacecraft used this mechanism to secure and subsequently deploy the despun section.

Array size is optimized to provide at least 8.25 kW during worst-case seasonal conditions and at the EOL of the solar cells. A possible candidate may be the EOS AM spacecraft, for which TRW is providing the solar array. Other designs are possible and available, e.g., two Lockheed Milstar arrays of silicon cells at 4,0 kW each and, possibly, Able Engineering's PUMA system. The deployed array will produce gravity torques to be managed by the ADCS three-axis system.

Given this arrangement, propulsion and attitude control are located at the other end. Full control authority can be achieved by locating a monoprop thruster system distributed at the four corners of the antenna structure, Twelve redundant thrusters provide full control in pitch, yaw, and roll. The directions are pure couples, providing good accuracy with regard to nodal crossings for the interferometric phases.

One of the 4 22-N engines is colocated in a cluster with the 3 5-N altitude control engines. The 22-N engines are required for orbital sustenance and maneuvers,

The only integrated design of the propulsion system is the tanks, which are to be integrated with the ACS. There are 5 fully Shuttle-qualified TDRSS monopropellant tanks. The total propellant capacity is 1530 kg.

A. Avionics

Electronics supporting the SIR-C mission are located on pallets on the ACS, arranged similarly to the SIR-C mission. The antenna core must support these boxes structurally, using additional members for strength. Thermal control is maintained passively by heat pumps connected to radiating pallets on either side of the structure.

The ACS is spacious enough to mount the electronics on modular pallets for easy integration. These pallets are mounted nearest the ACS center of gravity and arranged for easy flight cabling, The spacecraft avionic boxes are similarly located on pallets (see ACS box layout, Section IV).

The new boxes associated with the spacecraft are the Command and Data Subsystem (CDS), Data Routing and Management Equipment (DRME), Power Distribution System (PDS), Attitude Determination and Control System (ADCS), and Telecom system. A complete functional layout is shown in Figure 3-1. Each of these subsystems will be described briefly.

The central spacecraft computer is derived from the Mars Environmental Survey (MESUR) Attitude, Information, and Measurement Subsystem (AIMS) design, It is based on the IBM 6000, although for this application, it can run at a much reduced clock rate. The architecture of AIMS assumes a Cassini design, using a VME bus and 1553 interfaces. If the interface is an internal card off the bus, it is referred to as an Engineering Unit (EU); a remote EU is an REU.

Figure 3-1 shows the central computer, with internal cards allocated to primary interfaces such as the Payload Data Interleave (PDI) telemetry and Payload Signal Processor (PSP) commands. (These are standard interfaces to the Shuttle retained for cost purposes.)

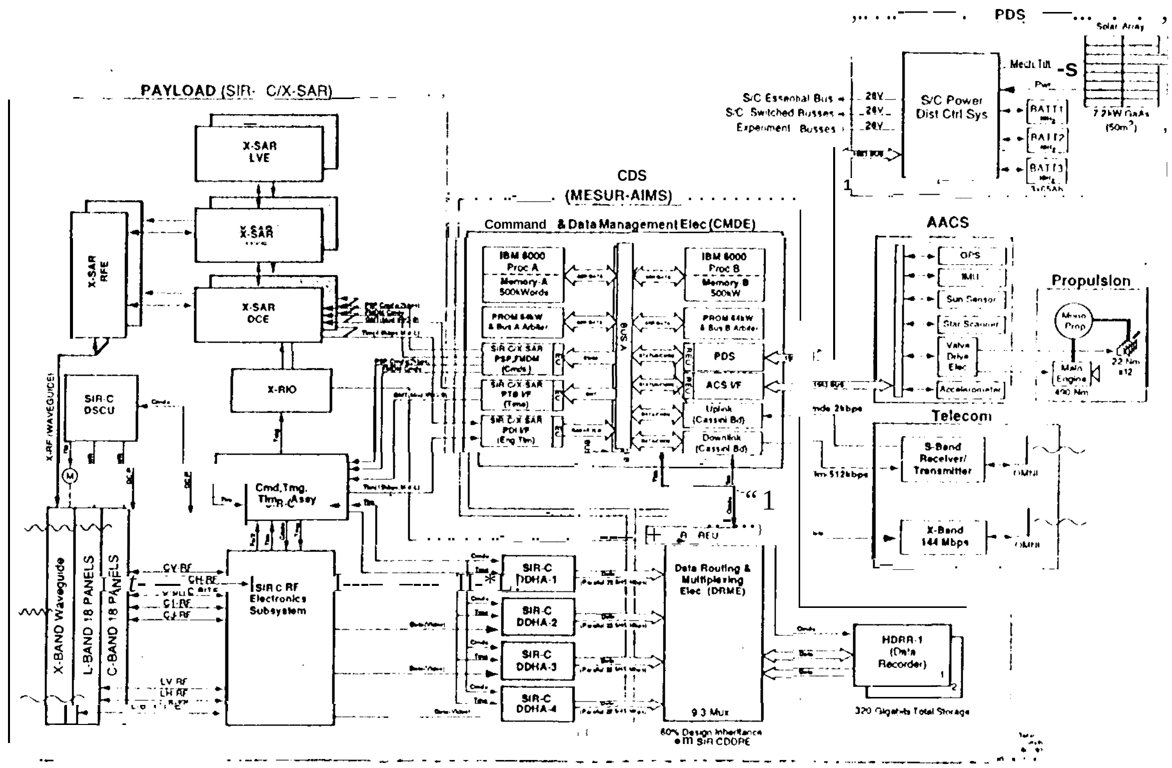

Figure 3-1. Functional Block Diagram

Uplink and downlink boards are available for command and telemetry through the S-band Telecom system. High-rate data are controlled by an REU supported by the DRME interfacing with two solid-state recorders,

There are two separate telecommunication systems: S- and X-band. The S-band system supports a 2-kbps command rate, based on an omni antenna and a 5-W transponder, The X-band system also has a 5-W transponder and an onmi antenna; it supports a downlink rate of 144 Mbps taking full advantage of the symbol rate associated with a bit error rate of 10^{3}. It will also support a downlink format of 100 Mbps, compatible with the present European network,

The ADCS consists of two star trackers, a Sun sensor, inertial measurement units (IMUs), and interfaces with the central computer via an REU. The primary guidance algorithms are implemented in the C language on the IBM 6000. Therefore, the central computer does both the command and telemetry and the ADCS functions.

The PDS steps voltage down from 120 to 28 Vdc and distributes it around the spacecraft. The solar array is based on the EOS AM mission design. There are three nickel-hydrogen batteries of 22 cell units each, based on the EUTELSAT-2 missions.

IV. Configuration, Electronic Packaging, Mechanisms, Cabling, and Structure

A, Configuration

The SIR-C Free Flyer configuration, referred to hereafter as the "spacecraft," is shown in Figures 4-1 through 4-7.

The spacecraft has been configured to make maximum use of existing SRL hardware consistent with new mission requirements; the entire SIR-C/X-SAR has been configurationally preserved. Most SRL pallet-located electronics modules have been used "as is" and integrated with the antenna core structure, along with their associated cabling, where possible. The STS interface trunnion loca-t ions have been maintained. The spacecraft mass, including fuel and mass uncertainties, is 12,707 kg. Newly required hardware was added as modules and attached to the ACS.

The 8 .25-kW solar array has been placed forward of the existing radar array for maximum illumination and minimum drag. The array deploys along an axis parallel to the velocity vector and rotates ±90 deg about that axis. The plane of the deployed array is always edge-on or parallel to the velocity vector for minimum drag. (See Figs. 4-4 and 4-7 for the stowed and deployed **cases.)**

The propulsion subsystem attitude control and velocity change (AV) thrusters have been configured as clusters at four corners of the radar array. For easy accessibility, propellant isolation and control assembly components are mounted on a modular panel located at the extreme aft end. Five monopropellant tanks are arranged inside the antenna core structure. Two star trackers and one Sun sensor are near the aft end of the spacecraft for attitude determination (see Figs. 4-2 and 4-3).

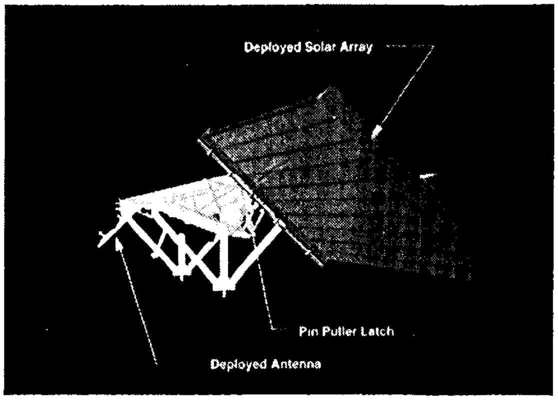

Figure 4-1. Deployed Spacecraft

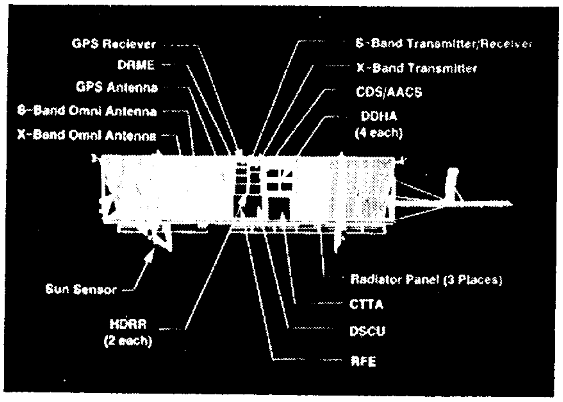

Figure 4-2, Aft View

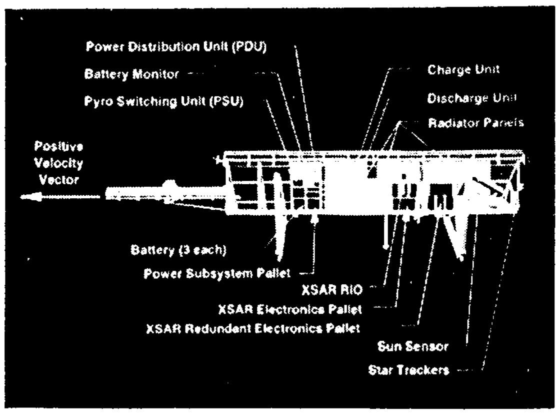

Figure 4-3. Port Side View

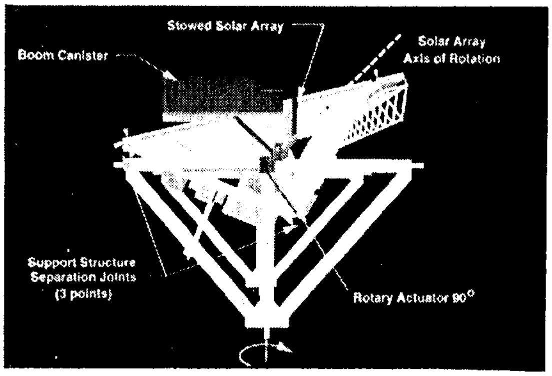

Figure 4-4. Forward View

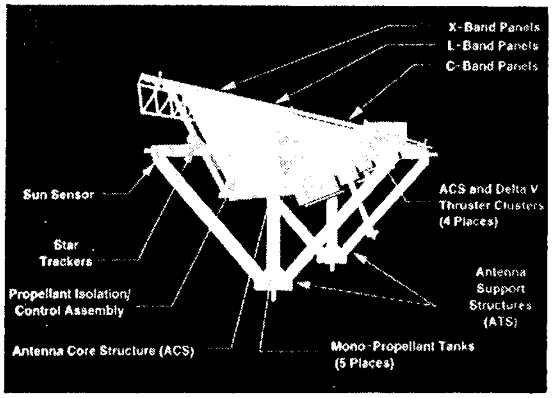

Figure 4-5. Starboard View

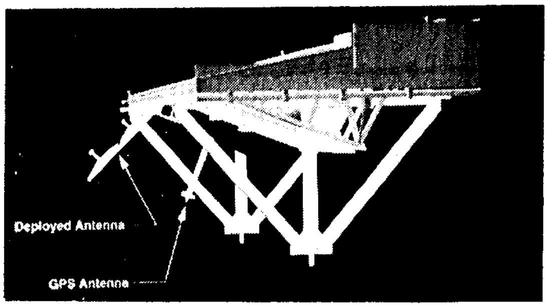

Figure 4-6. Deployed Antenna

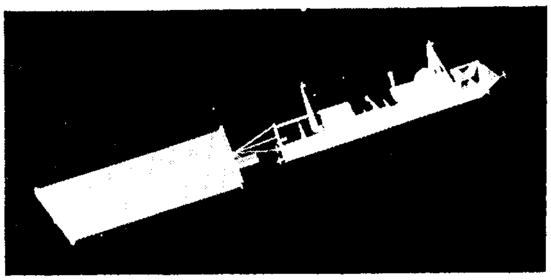

Figure 4-7. Solar Array, 0°

Two fixed nadir-pointing omni antennas (X- and S-band) are configured on a single mast, stowed and latched alongside of the core structure prior to separation from the STS. The mast is subsequently unlatched, deployed, and locked in the nadir-pointing position. These antennas are not steerable (see Figures 4-5 and 4-6). A fixed GPS antenna is mounted on the antinadir side of the spacecraft. Twelve 5-N thrusters are arranged in pure torque couples for pitch, yaw, and roll. Four 22-N thrusters are used for ΔV maneuvers; the forward pair are canted 45 deg to avoid plume impingement on the solar array.

All new hardware is integrated with the ACS. Six heat-pipe-fed thermal radiator panels are attached to the ACS adjacent to five electronics pallets. MLI blankets will encapsulate the radar antenna core structure and all attached subsystems. Thermal blankets have been omitted from all figures for clarity.

B. Electronics Packaging

All new electronics modules are currently packaged designs, except the ADCS/CDS; X- and S-band transponders will require new boxes. The approach has been to provide five separate, modular, structural pallets to the antenna core structure and then to arrange the electronics upon these pallets by subsystem, where possible. (See Figures 4-3 and 4-5 for the general arrangement.)

The existing and newly added but redundant X-SAR electronics have been arranged on two separate pallets located on the port side of the spacecraft as shown in Figure 4-3. The redundant electronics have been rotated on the pallet such that the wave guides exiting the primary and redundant X-SAR radio frequency equipment (RFE) are facing each other. This configuration allows an efficient and balanced distance of wave guide routing to the common radio frequency (RF) wave guide switch. The port side, aft, has been chosen for pallet location in order to minimize wave guide length from the switch to the existing X-SAR rotary joint. This will be significantly shorter than the existing SIR-C/X-SAR length. This approach should also reduce the required number of flexible waveguide segments. Since peak power dissipation is high, thermal control louvers or other radiative means may be required to dump excess heat.

The two X-SAR electronics pallets are totally modular and exclusive of other subsystems. The X-SAR REU package resides on the primary X-SAR electronics pallet. Various power subsystem electronics modules are arranged exclusively on one pallet located on the port side, forward, as shown in Figure 4-3.

The existing CTTA and four Digital Data Handling Assembly (DDHA) modules are mounted on a single starboard side pallet. The existing SIR-C RFE and Deploy/Stow Control Unit (DSCU) share a pallet with these newly added electronics: two High-Rate Digital Recorders (HRDRs), a

DRME, a ADCS/CDS, and S-band transmitter/ receiver, an X-band transmitter, and a Global Positioning Satellite (GPS) receiver. (Figure 4-5.)

All previously described modules attach directly to the aluminum honeycomb flat plate pallets. The pallets will be designed for handling and integrating the pallet field joint mate with the core structure. Pallets will be designed, where required, to maximize the heat path through heavy face sheets and the conductive core. An extensive use of heat pipes between electronics pallets and thermal radiators is used to maintain a constant 5-deg C, compensating for varying Sun angles. Heat pipe interfaces with pallets will be required.

C. Mechanical Devices

Spacecraft mechanisms, exclusive of those already integrated within the existing designs of the subsystem hardware, consist of the following:

(1) *Pyrotechnically Actuated Separation Nuts and Bolt Catchers.* Three 3/8-in .-diameter separation nuts and bolt catchers are used to preload the solar array support structure to the antenna core structure. They transmit launch loads and are actuated (separate) after spacecraft separation from STS, allowing subsequent engagement of the solar array clutch and rotary actuator. (Figs. 4-1, 4-4).

(2) *Clutch.* One each, two cones, with self-locking tapers, are separated longitudinally during launch environment to isolate loads from the rotary actuator. After separation nuts (item 1) are fired, four compressively loaded springs drive the cones into locked engagement, This moves the array and support structure longitudinally forward by - 1/2 in., allowing clearance for subsequent rotary motion of the solar array. This device is an existing design used on Galileo at the spin-despin rotary joint (see Fig. 4-1).

(3) *Solar Array Rotary Actuator* (Reference). This device resides in the attitude control subsystem (see Fig. 4-1).

(4) *Pyrotechnically Actuated Pin Pullers.* One 1 /4-in .-diameter pin device is used to latch a single deployable antenna mast during launch environment. Both S- and X-band omni antennas are mounted to the mast. Actuation of the pin puller allows the antenna mast rotary actuator to deploy the mast. A second pin puller latches the solar array strongback to the solar array support structure. This device is an existing design and has extensive flight history (see Figs. 4-5 and 4-6).

(5) *Antenna Mast Rotary Actuator.* Onc each of a damped, stored-energy, leaf spring device rotates the antenna mast from its stowed and latched position to its deployed and locked nadir-pointing position. A second rotary actuator rotates and locks the solar array and strongback 90 deg from launch position to the predeployment position. It is located at the interface between the solar array strongback and the support structure. Numerous devices of this type are available as existing designs with flight history (see Fig. 4-6).

D. Structure

The existing SIR-C/X-SAR instrument mass, exclusive of the SRL pallet-related mass, is approximately 7500 kg. The new spacecraft dry mass is approximately 11,200 kg. This represents an increase of 60 percent for rigid body inertial loads. With the exception of joints, the basic structural members of both the ACS and the ATS are capable of significant increases in loads. These judgments are based on conservative factors of safety used on SIR-C (2.5 to 2.7) and large positive margins on many, although not all, structural elements. Added reinforcements to the existing structural system will be required; however, the basic structure can be preserved. A finite element model of the new spacecraft will be required to specifically define which elements of the structure must be modified or reinforced.

We estimate 763 kg (1680 lb), exclusive of uncertainties, for these structural changes.

V. Power

A. General Description

The SIR-C Free Flyer electrical power subsystem provides a single regulated 28-Vdc power bus. Electrical energy is provided by a deployed photovoltaic array during sunlight operation and three 22-cell nickel-hydrogen batteries during eclipse operation or when load transients exceed solar array capability. Bus regulation is furnished by an array limiter unit. It holds the bus at a commanded set point, using quarter-volt increments. Depending on the load requirements, the duty cycle of the limiter switches and down converter are controlled to regulate bus voltage. Battery charge management is provided by the Power Distribution Subsystem (PDS), which uses absolute battery cell pressure and temperature as a means of monitoring battery charge. The PDS configures the battery charge unit to a high rate or trickle charges the batteries. Other electrical power subsystem functions include load current monitoring, load power switching, battery reconditioning, and cell voltage monitoring along with pyrotechnic event activation."

An internally redundant battery charge unit is provided for charging the batteries at one of two commanded rates: high-rate charging at C/20 (C amp-hours/hours) and trickle charging at C/200. The unit is operated in standby redundant mode so only one-half is active at any time. The CDS monitors battery cell pressure and temperature, comparing the data to programmable levels, Depending on whether data are below or above set point levels determines the battery charge. The CDS control signal to the charge unit may be overridden by ground command.

Battery voltage is converted to regulated bus voltage by a discharge unit. Control circuits in the discharge unit sense the bus voltage and compare it to a reference level, indicating excess solar array power is available, Signals are sent to the charge unit to prevent simultaneous battery charge and discharge, In the event of a discharge control signal failure, the failed discharge control circuit can be disabled, thus restoring normal charge unit operation, Additional discharge unit functions include discharge current and spacecraft bus voltage monitoring.

The battery monitor unit serially samples cell voltages and provides information on a telemetry channel. These data will be used during battery charge, discharge, and reconditioning to ensure that individual cells are not reversed or overcharged.

Distribution unit power-switch assemblies and fuse networks protect the spacecraft bus and switch bus loads. A power-switch assembly allows reconditioning by connecting a C/l 00 load to the battery upon command.

A capacitive-discharge bridgewire pyro drive unit is supplied that can activate 16 squibs. A maximum of four pyrotechnic devices can be fired simultaneously, using the three-command sequence of enable, arm, and fire. The unit is intended to release solar-array tie-down bolts and activate the propulsion system.

B. Power

The power distribution system is shown in Figure 5-1. Beginning with the solar array, 28 V of power is supplied to the instrument and avionics. The power requirements were derived from the worst-case orbit in the edge-on Sun exposure.

C. Solar Photovoltaic Array System

The baseline solar array concept described here for the SIR-C Free Flyer is a derivative of the

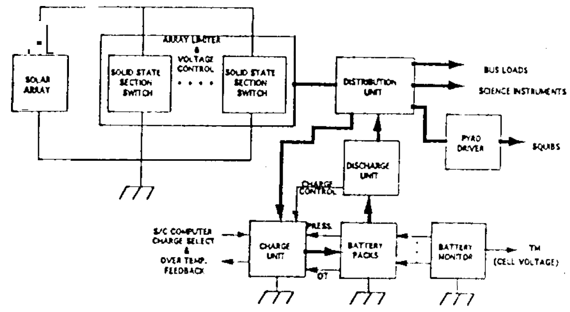

Figure 5-1. Power Block Diagram

EOS-AM solar array presently under development at TRW. That array is, in turn, a derivative of the NASA/JPL-developed Advanced Photovoltaic Solar Array (APSA), an ultralight-weight flexible blanket fold-out design. Since the SIR-C spacecraft bus was originally designed for use with Shuttle-provided power, it is necessary to retrofit it with a solar array system for the Free Flyer configurate ion. In view of the high power levels required (now in excess of 8.25 kW) and the limited stowage volume, a flexible fold-out design was felt to offer significant advantages in packaging. The TRW EOS-AM array seems to satisfy preliminary SIR-C power and configuration requirements. In addition, it has a number of other advantages, such as (1) being currently fabricated for a mission, (2) using GaAs/Ge (gallium arsenide/germanium) solar cells of over 18-percent efficiency, which minimize the deployed array area, and (3) being designed for atomic oxygen survivability in the Low Earth Orbit (LEO) environment.

The deployed SIR-C array will be 11 .8-m long and 5.1 -m wide for a total area of 60.2 m^2 (see Figure 5-2). A total of 44,640 190-pm thick GaAs/Ge solar cells are used to generate the array power. Each cell is 2.4 by 4 cm^2 in area and covered with a 150-μm thick coverglass; 190 cells are connected in series to allow a bus operating voltage of 127 V at end of life (7.5 years for EOS-AM). The folded blanket is deployed by means of a 14-in .-diameter coilable longeron mast system, Blanket location during deployment is constrained by four guidewires that attach at discrete positions on the array rear surface. When fully deployed, the blanket is tensioned to 60 lb by means of a series of constant negator springs. The semiflexible blanket material consists of a laminate of Ge-coated Kapton/graphite fabric/Ge-coated Kapton. The germanium coating provides atomic oxygen protection to the blanket assembly. The array wing will employ a single-axis rotary joint to follow the yearly variation in Sun angle. Daily solar angle changes will not be tracked. The array mass is projected to be approximately 207 kg.

D. Battery Description

Three 22-cell nickel-hydrogen batteries provide secondary power to the bus through the battery discharge unit. The battery packs are connected in parallel. During normal sunlight operations, the battery charge is maintained on a continuous trickle (C/200–C/100 rate). Total battery capacity is designed to provide power for all spacecraft loads during eclipse and supplement the solar array during peak loads. Following an eclipse or heavy drain, the batteries are recharged at a C/2–C/20 rate. The PDS computer will control battery charging by configuring the charge unit based on cell pressure and temperature. It is expected that the end of charge pressure and temperature limit will require calibration once each eclipse season, but otherwise will be fully automatic. Periodic reconditioning should not be required for a two-year mission; however, a resistive load is provided to discharge the batteries at a C/100 rate if desired.

The average discharge battery voltage is based on an average cell voltage of 1.26 V at beginning of life, and 1.24 V at 10 years in orbit (if necessary). The Depth of Discharge (DOD) of the batteries is based on a capacity of 195 ampere-hours. Normal DOD is expected to be 50 percent. The batteries can support all (nonmapping) spacecraft loads during eclipse with one cell failed at less than 60 percent DOD, and with a DOD less than 80 percent with up to three cell failures. The batteries state of charge can be determined on the ground by telemetry along with individual cell voltages obtained from the battery monitor unit.

VI. Propulsion

The Free Flyer propulsion system is the simplest and most reliable known. We selected a monopropellant, single blow-down system, fully redundant at the thruster level. Components of this design are flight proven and exist as production items today. See Figure 6-1 for a schematic diagram. There are 5 tanks, 4 clusters of 5 each, and 22 thrusters, which are available from two U.S. sources.

The primary components are 5 monopropellant tanks, 12 5-N thrusters used for three-axis attitude control, and 4 22-N thrusters used for ΔV maneuvers. The balance of components are standard pressure transducers, pyro valves, latch valves, service valves, filters, and temperature sensors. The system function is a blowdown mode.

The 4 1.4-in.-diameter propellant tanks are STS qualified, have extensive history on TDDRSS spacecraft, meet the 306-kg fuel capacity requirement, have envelope and attachment compatibility, and are available on a relatively short schedule (12 to 16 months). Tank operation pressures during blowdown mode are compatible with engines.

The maximum continuous on-time for the 22-N ΔV thrusters is approximately 32 min. This corresponds to a maximum 5 m/s velocity change.

The 5-N thruster pitch, yaw, and roll system is operated nominally in the maximum impulse bit

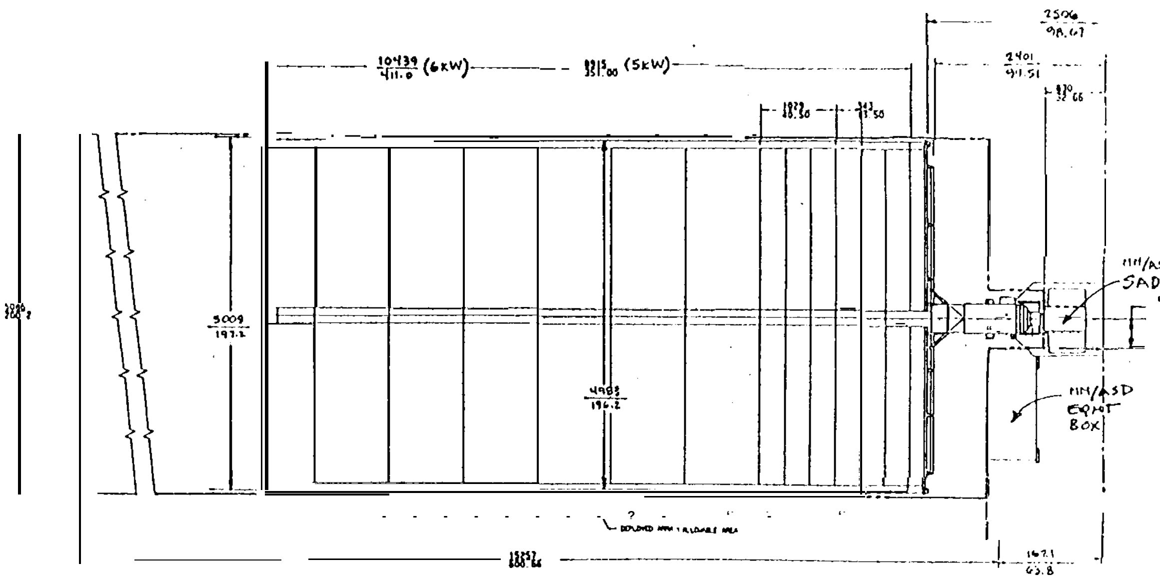

Figure 5-2. Solar Array Design

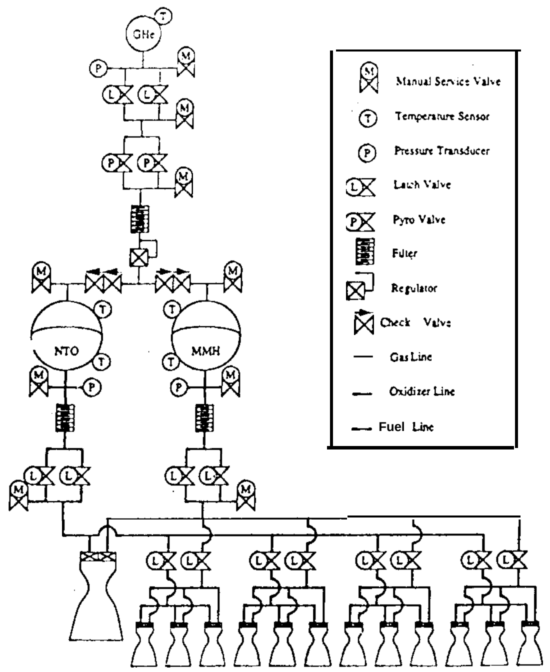

Figure 6-1. Free Flyer Propulsion Schematic

mode for attitude control; this system will also be used to point the spacecraft for AV maneuvers.

Propellant tanks and internal fuel feed lines must be installed early in the system integration phase. Thruster clusters and propellant isolation assembly/pressurant control assembly (PIA/PCA) modules may be integrated later due to their external mounting. Field joints in propellant lines will be very effective in minimizing cost and schedule.

VIII. Guidance and Control

A. Functional Requirements

The SIR-C Free Flyer ADCS is being designed to satisfy the following functional requirements:

- Initially acquire attitude and perform maneuvers during separation from the STS cargo bay.
- Precisely control the magnitude and direction of thrust impulses to achieve AVS,
- Perform rotation maneuvers to point the spacecraft in the required directions during AVS.
- Maintain precise attitude determination and pointing control during science observations.
- Monitor for component anomalies and autonomously compensate or enter safing mode.
- Rotate the solar array to achieve the optimal angle with respect to the Sun.

In addition to the functional requirements, the ADCS was designed with objectives of simplicity, robustness, functional redundancy, and low cost. Mass and power consumption, usually great concerns in the design of interplanetary spacecraft, are not critical on the SIR-C Free Flyer owing to low-altitude orbit and the large amount of power available from the solar panels,

B. Performance Requirements

During science observations, the ADCS must provide precise pointing and spacecraft control relative to nadir. These 3-u science control requirements include: 0.1 deg in pitch and yaw, and 1.0 deg in roll. A preliminary error budget allocates the majority of this allocation to control, allowing a maximum deadband width to minimize fuel usage during the two-year mission, To counteract a Doppler effect caused by the relative velocities of the spacecraft and ground, a sinusoidal ± 2-deg yaw will be applied to the spacecraft each orbit. During the interferometry phase of the mission, the spacecraft velocity must be controlled to within 1 cm/s. To achieve this requires a small AV along the velocity direction approximately once per day to compensate for atmospheric drag. Spacecraft velocity is determined by position measurements provided by a GPS receiver and from ephemeris data uplinked from the ground.

The interferometry phase also requires multiple AVS in directions normal to the orbital velocity vector. These AVS will be preceded by rotations of the spacecraft through either 45 or 90 deg. To complete the AV within a single orbit, the rotations should be completed within one-quarter orbit. Smaller "cleanup" burns will be required during two additional orbits, but the attitude control thrusters will be used for these burns to avoid rotation maneuvers.

The incidence angle between the Sun and the spacecraft changes according to its position in the orbit, and also according to the gradual precession of the 57-deg inclination orbit. To maximize the average illumination of the solar array for maximum power output, the array can be rotated ± 90 deg about the spacecraft roll axis. Adjustment of the solar array roll angle will be made on a daily basis, rather than continuously, which introduces disturbances to the system, complicates control algorithms, and subjects the array drive actuator to an excessive duty cycle.

C0 ADCS Hardware Description

Monopropellant thrusters with a rated output of 5 N will be used for a three-axis attitude control. Their locations are selected so that pure torques can be created about the yaw, pitch and roll axes of the spacecraft by firing pairs of thrusters. The configuration can accommodate failure of any single thruster with no loss of functionality. For calibrations and other special functions. The computer must have the capability to perform these functions in a timely manner and to respond quickly to changing conditions, Due to the large size of the spacecraft, the required bandwidth of commands is not expected to be very large relative to other spacecraft.

D. Fuel Use Calculations

The specific impulse of monopropellant thrusters ranges from more than 210 s during steady state operation to less than 100 s for short duration (<20 ms) pulses, and is also dependent on duty cycle. The thrust level also varies slightly with pulse width and duty cycle. Hence, calculating fuel usage is an iterative process because the pulse width and duty cycle cannot be known unless the thruster performance is known.

Using a 5-deg principal axis misalignment, the results of fuel use calculations are summarized in Table 8-1.

The parasitic fuel loss due to gravity gradient torque can be reduced significantly by one of two approaches: either reduce the level of gravity gradient torque itself (5 deg is very conservative and contains propellant margin), or add an additional subsystem to augment the monopropellant thrusters. To reduce the torque, the spacecraft must be carefully balanced such that principal axes align with nadir. This must be done to the extent possible by careful arrangement of the electronic boxes on the spacecraft, and possibly through the use of ballast. With a perfectly balanced spacecraft, gravity gradient torque will still occur about the roll axis due to rotation of the solar panels, but this is minor compared to that about the pitch axis.

Two alternative approaches are also suggested to reduce the gravity gradient loss. One approach emphasizes a single reaction wheel aligned with the spacecraft pitch axis. Momentum unloading would be achieved using magnetic torque rods. The torque generated by a torque rod is perpendicular to the local magnetic field; hence, unloading can occur only near specific points in the orbit.

Table 8-1. Fuel Use Summary

Source	2-Year Fuel Use
Gravity Gradient	664 kg
A-V Maneuvers	351 kg
AV Maneuvers	306 kg
Atmospheric Drag Force	62 kg
Solar Pressure Torque	50 kg
Periodic 2-deg Yaw	40 kg
Rotations for AV	40 kg
Atmospheric Drag Torque	6 kg
Deadband Control	2 kg
Total	1215 kg

The other approach employs a single 30-cm ion engine in a moment-based control configuration to augment the monopropellant thrusters. Its primary advantage is a far higher specific impulse of -1760 s, as opposed to -150 s for the monopropellant thrusters in their typical duty cycle.

IX. Major Options

Integration of spacecraft subsystems onto an existing structure, constrained by the shuttle environment, is a unique challenge. The instrument's external environment (shuttle) is completely replaced. Subsystems now provide command, attitude control, and power. The active cooling system of the shuttle pallet is replaced by a unique passive thermal system, using heat pipes.

The baseline mission of a multifrequency, multiphase, fully polarized observation is extremely attractive since spacecraft additions are only about $100 M, to be shared with European partners.

X. Acknowledgment

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